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FEASIBILITY STUDY FOR A COMPACT ENVIRONMENTAL ANOMALY SENSOR (CEASE)

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13. ABSTRACT *(Maximum 200 words)*
The local environmental conditions that a spacecraft operates within may induce occasional anomalous behavior that can impact reliability and performance. The Compact Environmental Anomaly Sensor (CEASE) is an instrument being developed to detect local environmental perturbations and provide the host spacecraft with various "anomaly" alerts along with confidence factors that allow the host spacecraft to conduct its operations in a manner to minimize these anomalous responses or to flag data as suspect due to local conditions. This report covers the first year of a feasibility study of this instrument.

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Introduction

During the first year, Amptek and its subcontractor SAIC have performed studies on the space radiation environment, spaceflight radiation monitoring hardware, and software algorithms for potential hazard assessment in order to determine practical scenarios for a successful CEASE instrument.

Several computer based models of the solar plasma environment were obtained. An effort was made to identify or create a data base of reported anomalies and precipitating conditions. We have found that there has been no consistent format for identifying environmentally induced anomalies. Furthermore, there has been no consistent plasma diagnostics to report conditions when such anomalies have been observed. This lack of anomaly identification combined with the variety of diagnostic capabilities and interpretations makes the gathering of such a data base difficult and of debatable value. We therefore do not anticipate pursuing developing such a data base. The CRRES satellite is uniquely suited to provide the data we need using a common and very comprehensive set of plasma diagnostics and covering a variety of solar environments.

The hardware study has concentrated on three aspects of the CEASE instrument challenge. We have tried to identify some candidate diagnostics and their size, weight, and power requirements. We have made an estimate of the probable volume of the CEASE and have identified radiation-hardened electronic components for the assembly.

The anomaly probability and risk assessment software will depend somewhat upon the suite of diagnostics that are selected. We have, however, studied the processing of real time plasma diagnostic data and looked at some strategies to accumulate and compress the data to forms that an anomaly hazard assessment program can rapidly scan and derive risk factors and a confidence estimate.

The CEASE contract was modified early in its first year. This modification eliminated most of the anomaly and diagnostics research that was originally proposed. The successful launch and flight of the CRRES satellite has provided much of the necessary data. Phillips Laboratory personnel will review and reduce the CRRES data and develop a list of recommended sensors and measurements that the CEASE instrument can use to monitor the space environment for conditions that may trigger satellite system anomalies. Amptek and SAIC will then develop and study the CEASE instrument using the suggested sensor suite and measurement ranges.

CEASE Hardware Possibilities

To establish a probable volume and thereby a shape factor for CEASE we have tabulated below the dimensions and weights of some recently completed spaceflight experiments.

Assembly	Weight (pounds)	Approximate Size (inches)	Density (lbs/in^3)	Density (gm/cm^3)
Data Recorder	18	6 x 9 x 11.8	0.028	0.775
SPREE ESA	4.7	10 x 6 x 6	0.0131	0.363
Rotary Table	13.2	10.5 diameter x 3.7	0.0206	0.570
SPREE DPU	19.3	12 x 10 x 6.9	0.0234	0.648
TED	9.9	5.7 x 12 x 6.8	0.0214	0.592
SSJ4	5.0	6 x 5.6 x 5.25	0.028	0.775

Table 1. Densities of some recent Spaceflight Instruments

This gives an average density of 0.0224 lbs/in^3 (0.620 gm/cm^3) as compared to solid aluminum which has a density of 0.098 lbs/in^3. A starting premise for CEASE is that the instrument will be limited to 6.8 kg (15 pounds) and 10 Watts power. The CEASE instrument will consist of some environmental sensors and data processing abilities. Based on the above data, it seems reasonable to assume that CEASE will have a density around .025 lbs/in^3 (0.692 gm/cm^3), which results in a volume of ~600 in^3 (9,827 cm^3). This is a relatively large volume; for example a box 10" x 10" by 6" high. In terms of adaptability to many spacecraft, a smaller volume would be desirable, and Amptek feels that a smaller and lighter implementation of the CEASE instrument is possible.

The CEASE instrument will require a microprocessor to interpret the data from the diagnostic suite and provide the host spacecraft with a system of weighted probabilities of imminent anomaly occurrence. The microprocessor will also provide a "personality" that can be customized to the host requirements. It will also be useful during development and testing stages of CEASE. Amptek has focussed on three possible candidates:

Microprocessor	Bus	Speed	Power	Total Dose (Krads)	SEU (Errors/bit-day)
80C85 (SA3000)	8 bit	5 MHz	5 mA/MHz	10,000	10^{-10}
80C86	16 bit	8 MHz	12 mA/MHz	1,000	10^{-2}
1750A	16 bit	25 MHz	TBD	1,000	10^{-6} - 10^{-12}

Table 2. Candidates for the CEASE microprocessor

The 80C85 has the highest reliability history for spaceflight, but it may not have sufficient speed and/or capability. The 1750A processor system is being examined as an alternative if the 80C85 will not suffice.

CEASE could have a number of possible diagnostics. Table 3 tabulates characteristics of some probable candidates:

DEVICE	FUNCTION	SIZE	WEIGHT	POWER
MOSFETS	cumulative radiation dosage	1.5" x0.5" x0.75"	0.05 pounds	0.02 Watts
SEU I.C.s	upsets	1.5" x1.5" x0.75"	0.05 pounds	0.02 Watts
Solid State Detector (SSD)	ionizing radiation	3" Dia, 4" long	1.2 pounds	0.3 Watts
Temperature-controlled Quartz Crystal Microbalance (TQCM)	cumulative mass deposited on a sample area	1" Dia. , 3" long	0.265 pounds (120 g)	0.35 Watts
Thermal Coating Calorimeter (TCC)	temperature of an isolated body in radiative thermal contact with space	1.25" Dia, 1"long	0.11 pounds (50 g)	0.1 Watts
Electrostatic Analyzer (ESA)	charged particle populations	4"x4"x4"	1.3 pounds	0.4 Watts
SPM	surface potential monitor	4"x5"x1"	0.5 pounds	0.02 Watts
TPM	transient pulse monitor	0.25"x1"x1"	0.02 pounds	0.02Watts
Sun Sensor	solar exposure	0.5" Dia, 1" long	0.05 pounds	0.01 Watts
Optical Scattering Sensor	bi-directional distribution function	1.25" Dia., 2" long	0.08 pounds	0.3 Watts

Table 3. Physical Characteristics of Diagnostics

A principal concern for the CEASE instrument will be radiation tolerance since it will need to reliably forecast impending disruptive levels of radiation while being designed to tolerate relative high dosages itself. In particular, the digital processor needs to be rad-hard and capable of fault detection and correction so that it can produce dependable results.

Rad-hard digital circuitry is becoming more common; however, it is necessary to select not only from a radiation hardness criteria, but also from a power dissipation and environmental criteria. A trade magazine, Military & Aerospace Electronic (Sept 1991), has recently published a compilation of manufacturers and suppliers of radiation-tolerant integrated circuitry which we have adapted for tabulation below in Table 4. This gives some indication of the range of suppliers and technologies that offer radiation hardened devices.

Company	Technology	Product Description	Total Dose (Krads)	SEUs (errors/bit-day)	Trans. Threshold (MeV/mg/cm^2)	Qualification
ABB HAFO	1.9-µ CMOS/SOS	SRAM	> 100	< 1 x 10^{-9}	> 43	ESA Cert (Pending)
		Counters, Registers, Bus Controllers	100 to 1,000	< 1 x 10^{-9}	> 43	ESA Cert
		ASIC - mixed signal	> 100	< 1 x 10^{-9}	> 43	ESA Cert
AMD	0.8-µ Bipolar	ASIC PLD	1,000	N/A	N/A	MS-883C
Harris Semi	1.2-µm CMOS/SOS	SRAM	> 10,000	< 1 x 10^{-12}	> 257	MS-883C Class S
	1.25-µm CMOS/SOS	µprocessors, standard logic	> 10,000	< 10^{-10}	> 120	
	1.2-µm CMOS/SOS	Gate Array, standard cell, Silicon Compiler	> 1,000	< 1 x 10^{-10}	N/A	
Honeywell	0.7-µm CMOS	SRAM	> 2,000	< 1 x 10^{-10}	115, SEU latchup immune	MS-883C Class S, QML Cert
	1.25-µm CMOS	Standard Cell, full Custom	1,000	< 1 x 10-9	SEU latchup immune	
	0.7-µm CMOS	Gate Array	2,000	< 1 x 10^{-11}	SEU latchup immune	
IBM Fed Div	0.8-1.0-µm CMOS	SRAM	> 200	< 1 x 10^{-10}	60-80	MS-883C Class S, QML Cert
	0.5-1.0-µm CMOS	Gate Array, standard cell	> 2,000	< 1 x 10^{-10}	60-80	
IDT	0.45-0.9-µm CMOS & BiCMOS	SRAM, dual port RAM	20 to 70	N/A	N/A	MS-883C Class S
		FCT logic, RISC µprocessors, FIFOs, etc.	30 to > 100	N/A	N/A	
Linear Tech.	10-µm Bipolar	Op-amps	200	N/A	N/A	MS-883C Class S
LSI	0.7-µm HCMOS	Gate Array	3,000	2 x 10^{-10}	52	MS-883C Class B
Marconi	1.2-µm SOS	SRAM	1,000	4.3 x 10^{-11}	59	MS-883C Class S
	1.2-2.0-µm SOS	1750 µprocessors, 29XX, peripherals	1,000	< 1 x 10^{-12}	180	
	1.2-µm SOS	Gate array, standard cell, custom	1,000	< 1 x 10^{-12}	> 180	

Company	Technology	Product Description	Total Dose (Krads)	SEUs (errors/bit-day)	Trans. Threshold (MeV/mg/cm^2)	Qualification
Micrel Semi	8-μm CMOS metal gate	standard logic	1,000	N/A	N/A	TBD
Micron Tech	0.7-0.75-μm CMOS	SRAM	> 10, > 30	N/A	1.5, 2	MS-883, some JAN
National Semi	1.25-μm CMOS	standard logic	100	$< 1 \times 10^{-4}$	> 40	MS-883C Class S, JAN
Raytheon	2-μm Bipolar	PROM	> 10,000	N/A	N/A	MS-883C Class S
Signetics	1.5-2.5-μm Bipolar	PROM, SRAM	10,000	N/A	N/A	TBD
	0.8-1.5-μm CMOS	μcontrollers	2,500	N/A	N/A	
Silicon Gen	Bipolar, CMOS, SOI	Pwr REg, pulse-width modulators	3,000	N/A	1×10^{13} (???)	MS-883C Class S, SMD
Sipex	4 to 8-μm Bipolar	OP Amps	1,000	1.2×10^{-8}	N/A	MS-883C Class S
	4 to 8-μm NPN/PNP bipolar ASIC	custom & analog arrays	1,000	1.2×10^{-8}	N/A	
TI	1 -μm CMOS/SOI	SRAM	200	$< 1 \times 10^{-11}$	75	MS-883C Class S
	3 -μm MOS	standard logic, HC/HCT	20	N/A	N/A	JAN Class S
	1 -μm CMOS/SOI	Gate Array	1,000	$< 1 \times 10^{-11}$	75	MS-883C Class S
TRW	1.5-μm Bipolar	custom	>3000	N/A	N/A	MS-883C Class S
UTMC	1 -μm CMOS	SRAM, masked ROM	1,000	1×10^{-10}	47	UTMC Level S
	1.25 -μm CMOS	RISC, 1750A, DSP	1,000	2.6×10^{-5} 1×10^{-8}	40	
	1 -μm CMOS	gate array	1,000	N/A	$<6.25 \times 10^{-8}$	MS-883C Class S
Vitesse	0.8-μm GaAs	SRAM, gate arrays	100,000	N/A	N/A	MS-883C Class B

Table 4. Rad-Hard Integrated Circuit Suppliers
Adapted from Military & Aerospace Electronics (Sept 1991)
Sentry Publishing Company, Inc.

Development of CEASE Operating Algorithms

During this first year of the program SAIC has attempted to determine what data sets are available that relate environmental conditions to anomalies, and whether the data are useful for CEASE. We looked into other experiments that have measured similar conditions and we have also thought about whether there are significant effects to be measured that were not named in the Statement of Work.

What we found in our data base and literature research was a spacecraft anomaly data base (SAM from the World Data Center A/National Geophysical Data Center) which tabulates anomalies without including space environment information. We also found large data bases which contained detailed time histories of the space environment but did not correlate anomalies. Finally, we encountered a number of restricted or classified data bases from which we obtained limited descriptions of diagnostic capabilities and observed anomaly thresholds.

RESOURCE	ANOMALY DATA	ENVIRONMENTAL DATA	INDIVIDUAL EVENTS
World Data Center/ National Geophysical Data Center	SAM data base TIROS in polar sun-synchronous orbit at 88 km. GOES in GEO orbit TDRSS	Time Series on tape: TIROS protons 300-2500 keV, electrons 30-300 keV, also auroral energy particle data. GOES protons 4-800 MeV electrons >2 MeV, B field x-rays 1-8 angstroms.	Papers on Aug-Sept 1989 solar storm. >3000 anomalies noted during this period.
Los Alamos National Laboratories (LANL)	Classified.	protons 70-150 keV electrons 30-2000 keV .001-40 keV electrons & protons with pitch angle	No unclassified data published with respect to anomalies.
Global Weather Center (GWC)	Case studies of anomalies upon request GPS events have been analyzed. AF must request study.	NOAA satellites Global Position Satellite (GPS)	No events published.
Orbital Data Acquisition Program (ODAP)	Apparently no useful data.	Apparently no useful data.	No events published.
Air Force CRRES Satellite	Mission designed to correlate anomalies with space environment.		Papers are being presented currently.

Table 5. Satellite Anomaly and Environment Resources

As can be seen Table 5, the CRRES data set is the only data base that systematically couples space environment with anomaly detection and reporting. It has the most comprehensive diagnostics and an orbit that passes through several plasma regimes, reporting on a variety of environments. Even if the space environment and anomaly occurrences of other satellites are correlated, the resulting data base will be difficult to interpret since the types, accuracies, and ranges of diagnostics varies considerably from mi. on to mission. It would be expensive and inefficient to develop a self-consistent anomaly/space environment data base from archived data, and the result would probably not be as accurate or detailed as the CRRES data. Access to the CRRES results will substantially improve the CEASE design phase.

The goal of the CEASE project is to develop a generic instrument for Air Force satellites to measure the environment, evaluate the hazard to spacecraft systems, and issue warnings of the hazard.

We have prepared the list of parameters that CEASE should measure in order to warn of the anomalies. Those explicitly mentioned in the contract SOW we call baseline. Other significant anomalies we call additional. We base our recommendation on the best information available to us, but we do not at present have access to data from the Combined Release and Radiation Effects Satellite (CRRES), which was expressly designed to measure the anomaly producing environment in the magnetosphere. We describe the anomalies and their causes and list the measurements needed.

The baseline anomalies as listed in the SOW are surface charging, deep dielectric charging, total radiation dose, and single event upsets (SEU). To these we add body charging and contamination as additional causes of anomalous spacecraft operation. Before recommending measurements, we briefly review these anomalies.

A. Description of Anomalies

1. Charging

Body charging is the collection by the spacecraft chassis or frame of net electrical charge that puts it at a potential different from the local plasma. The potential drop occurs across a plasma sheath. Body charging occurs, even when the entire spacecraft *surface is* conducting and connected to chassis ground, whenever there is net current between the spacecraft and the background plasma. A potential difference develops to repel or attract charge until the net current becomes zero. Typical incoming currents are hot electrons or cold ions. Outgoing currents are photo- and secondary electrons. As the typical capacitance of a spacecraft with respect to space is 100's of picofarads, this adjustment typically occurs with microsecond time scale.

Fortunately, body charging typically does not damage a spacecraft or cause anomalous operation except to distort the energy spectrum of arriving particles. While this is of great importance in interpreting measurement of the natural environment for scientific purposes, it normally does not damage spacecraft. Body charging can produce two possible anomalies: if the spacecraft discharges or arcs to the plasma as from a sharp point there would be electrical noise that might disrupt vehicle operation. We believe that this has not occurred in the natural environment, but it may happen on vehicles that internally generate high potential differences between circuit elements, or that eject charge in energetic particle beams. *Olsen and Norwood* [1991] show that body charging can produce sputtering from the spacecraft from incoming ions accelerated by the spacecraft potential. Incoming particles can also enhance the other types of charging.

Surface charging occurs in exposed dielectrics bombarded by charged particles that penetrate several microns (5 or more) into the surface. If the leakage current through the dielectric is smaller than the incoming current, an electric field develops which may become very large over a short distance. Once the field reaches on the order of 10^6 V/m, the dielectric breaks down with electrostatic discharge (ESD) within the dielectric. Somewhat lower fields may discharge from edges to another surface. Bursts of X-rays or other ionizing radiation can trigger ESD. ESD causes problems by radiating noise into sensitive circuits or introducing sufficient voltage to damage components.

Surface charging requires slightly penetrating particles, occurs over many minutes or hours as the buried charge density builds up, and is flux dependent. Charge deposition is so shallow that secondary yield, backscatter, and photoemission affect surface charging. Sunlight produces copious low energy photoelectrons on the outside surface of the spacecraft. This cloud of electrons can decrease charging by neutralizing the current from incoming electrons, but it can also increase charging by changing the potential between two adjacent surfaces, one in light and one in shadow.

Deep dielectric charging is the same as surface charging except that the charges penetrates deep enough (>250 μm) that surface effects do not come into play. Particles have higher energy to penetrate to this depth. Charge may accumulate in dielectrics or insulated conductors such as traces on circuit boards. Deep dielectric charging may result even if the spacecraft skin is entirely conducting. As with surface charging there need be no net body potential, although such a potential accelerates some incoming particles to higher energy.

For both surface and deep dielectric charging in a planar geometry where the electric field is from a buried layer of charge to an outer surface, incoming fluence on the order of 10^{11} /cm^2 produces the required field of order 10^6 V/m. The necessary current (equivalently flux) depends upon the dielectric conductivity as does the buildup time, but no effects have been observed in the laboratory with currents < 0.3 pA/cm^2 [*Robinson*, 1989; *Beers*, 1988]. Radiation damage increases conductivity (see below) and reduces the ability of a dielectric to charge. Thus the entire distribution of particles is significant.

Differential charging occurs when different segments of the spacecraft experience different particle flux or sunlight.

In a sense there is no definite energy boundary between particles with range to cause surface charging and deep charging. All particles with range greater than some 5 μm can cause one or the other if the flux is high enough.

2. Radiation

Total Dose is the integrated amount of ionizing radiation. The damage occurs from the passage of penetrating charged particles as they lose energy by collisional ionization. Junctions in solid state devices are most sensitive, but at high enough accumulated dose, optical properties degrade and conductivity of dielectrics increases. This damage is permanent although some devices anneal to a degree when allowed to rest. The critical dose depends upon the device. Critical doses range from kilo- to megarads.

Single Event Upsets (SEUs) result when penetrating particles create by collisional ionization enough separated charge in the sensitive volume of a junction to alter the logical state of a circuit using the junction. The critical charge or *Linear Energy Transfer* (LET) varies from device to device. With present devices, only $Z>1$ particles have sufficient LET to cause an SEU. Energetic protons can produce a nuclear reaction with a silicon nucleus to produce multiple heavily ionizing fragments (nuclear stars). Although the cross-section for nuclear interactions is small, protons are much more numerous than $Z > 1$

particles in the radiation belts and in solar flare particles. Preliminary CRRES results are that protons are the major cause of SEU's.

3. Contamination

Contamination may be either molecular or particulate. Particulate solids or droplets in the field of view of instruments may scatter or absorb electromagnetic radiation. Particles that scatter sunlight into tracking devices may appear as spurious targets. Molecular (*i.e.*, gaseous) contamination may have plasma effects as a gas and may condense on portions of the spacecraft, especially cold ones. It is possible to measure the total condensed mass or the possible effects. Table 6 shows the critical thicknesses for some of these effects.

Sensor/Surface	Spectral Bands (μm)	Maximum Deposition (μm)
EUV - UV	0.12-0.40	0.001
VIS - near IR	0.40-2.20	0.01
IR	4.0-16.0	0.05
Thermal Radiator	NA	0.2

Table 6 Critical mass depositions for various sensors and surfaces

Optical effects

Condensation on optical surfaces scatters, absorbs, or reflects light. This degrades optical performance including that of solar cells.

Thermal effects

Condensation on thermal control surfaces such as radiators may alter their absorptivity or emissivity thus changing spacecraft temperature.

Electrical effects

Condensate on a conducting surface may be insulating, thus altering contact with the surrounding plasma and changing the charging environment.

B. Recommended Measurements

Table 7 summarizes the environments in the magnetosphere that can produce each of these effects. Most of the observed spacecraft anomalies are attributed to ESD resulting from surface or deep dielectric charging, or to SEU. Most anomalies have occurred in GEO. Degradation from total dose is seen in solar cell performance and in other systems. Contamination is known to have affected thermal performance. We list measurements by CEASE to warn of conditions that lead to significant anomalies. Of these, all but SEU are cumulative, so that there is time to warn of a threat. With SEU, the sensor will have great enough sensitive volume to warn of upsets when their rate in spacecraft systems is very low.

Anomaly	Cause: Flux & Energy	Where Found	Recommended Measurement
Body Charging	5-50 keV electrons. Worse during eclipse or when low energy electrons are absent.	GEO and Substorm. Energetic aurora with low plasma density.	None
Surface (Differential) Charging (3-60 μm of dielectric)	Fluence $>5.5\times10^{11}/cm^2$ Flux >1 pA/cm^2; 5-50 keV electrons, 0.1-1.5 MeV ions. Worse when sunlit or spin stabilized or absence of low energy electrons. Slow charging.	GEO and Substorm. Inner/Outer belt, L=1.2-4.	Electrons 5-50 keV. Protons 0.1-1.5 MeV. Use a flux threshold $2\times10^6/cm^2$-s. Presence of sunlight.
Internal Charging (250-13000 μm of dielectric or metal)	Fluence $>5.5\times10^{11}/cm^2$ Flux >0.3 pA/cm^2 ($2\times10^6/cm^2$). 0.3-7 MeV electrons, 6-55 MeV ions. Slow charging.	Outer Belt for ≤1 MeV, L=1.5-2.5.	Particles able to penetrate 250 and 10,000 μm Al. Separate protons and electrons. Flux threshold $2\times10^6/cm^2$-s.
Total Dose Degradation	Any ionizing penetrating radiation. >0.3 MeV electrons, >10 MeV protons, Cumulative.	Cosmic Rays in all orbits. Solar flare particles high lat. Inner/Outer belt, L=1.2-3.	Threshold Shift in biased p-n junction. Integrated measurement. Warnings at decade intervals starting at 1 krad.
Single Event Upsets (SEUs)	>10's MeV protons that produce nuclear reactions in silicon; $Z>1$ particles.	Cosmic Rays in all orbits. Solar flare particles high lat. Inner belt L=1.2-2. (In LEO, South Atlantic Anomaly is the major contribution.)	SEU rates in memories with varying LET thresholds to detect protons and $Z>1$. Threshold above Cosmic Ray.
Mass Contamination	Emissions/degassing and condensation on spacecraft with deposition $>10^{-7}$ g/cm^2.	S/C generated.	Cumulative Mass Deposition: 0.5×10^{-7}, 0.5×10^{-6}, and 2.5×10^{-6} g/cm^2. Thermal properties shift.

Table 7. Causes of Anomalies

Strawman CEASE Instrument

CEASE Charter

Our charter is to develop a generic instrument for Air Force satellites to measure the environment, evaluate the hazard to spacecraft systems, and issue warnings of the hazard. The initial hazards are: total radiation dose, single event upsets (SEU's), dielectric surface charging, and deep dielectric charging. The package guidelines are 15 lbs, 10 W, in a single package.

Overview

Two points have become evident while functionally designing the CEASE package. First, depending on orbit parameters and type of payload (e.g. scientific vs. communication), spacecraft needs differ considerably. Second, some sensors are sufficiently compact and lightweight to be co-located with the equipment for which the hazard is to be evaluated. Consequently, our concept for the CEASE package (Figure 1) differs from the charter corresponding to these points. Further, we have added the option of contamination sensors.

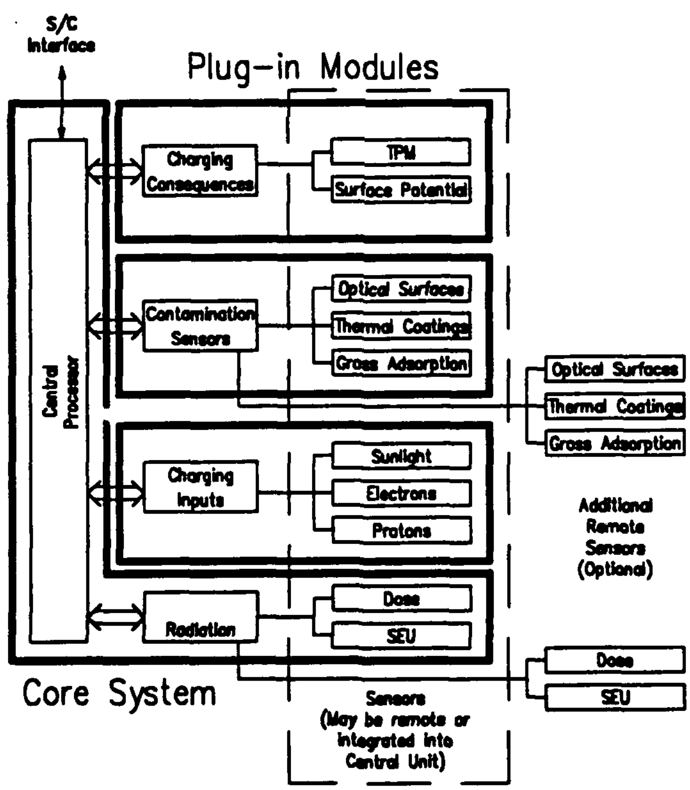

Figure 1. CEASE Strawman Concept.
The sensors may all be located in one box or distributed as convenient.

1. Modular Central Unit

The central unit is modular and contains only those systems that are appropriate for the spacecraft upon which it is installed. This allows the package to be more compact for most installations than the charter calls for. The central unit has the core processor and interface to the spacecraft, radiation sensors, and "slots" for plug-in modules for charging input, contamination, and charging consequences. Each module has a sensor and an interface circuit. Since for non-polar LEO, charging is not a concern (except for active charge-emitting satellites), LEO spacecraft need not include either charging module. The contamination module is of most interest for satellites with optics and satellites with long lifetimes that are concerned with the degradation of thermal surfaces.

The processor would handle all the possible modules. The most basic package is the processor and radiation sensors.

2. Optional Remote Sensors

Remote sensors can be plugged into the base unit to allow monitoring of the hazard to selected parts of the spacecraft. This applies to radiation and contamination monitors. Further, some of the sensors could be remotely located for convenience. It makes integration more difficult, but may make fitting in the pieces easier.

Detector Description

Table 8 summarizes the modules, sensors and their required volume.

Module	Sensors	Sensor Dimensions: Outside Area (cm)	Sensor Dimensions: Thickness (cm)
Radiation	MOSFETS	4 x 1	2
	SEU IC's	4 x 4	2
Charging Inputs	Sunlight Sensor	1 x 1	1
	GM Tube	1 Dia	7
	Solid-State Telescope	10 x 10	10
	Electrostatic Analyzer	10 x 10	10
Contamination	Quartz Crystal Microbalance	2.5 Dia	7
	Thermal Coating Calorimeter	3 Dia	2.5
	Bidirectional Reflective Distribution Function	3 Dia	5
Charging Consequences	SPM	10 x 10	10
	TPM	1 x 6	1

Table 8. Cease Sensor Summary.
Dimensions are representative for actual sensors only and do not include supporting electronics.

1. Radiation

The core CEASE sensor is for radiation dose and SEUs.

The total radiation dose can be measured by looking at voltage threshold shifts in MOSFET devices. Len Adams of European Space Agency (ESA) has developed specialized Field Effects Transistors (FETs) for this purpose called RADFETs that can be manufactured with various sensitivities [*Adams et al.*, 1991]. Martin Buehler of Jet Propulsion Laboratory (JPL) constructed application specific integrated circuits (ASICs) for CRRES that had 32 MOSFETs in them. These monitored radiation dose behind various shielding thicknesses during flight. Several of these would be located in the core unit. Since these devices are quite compact and light, additional detectors can be located remotely near devices for which radiation is a particular concern.

SEUs can be predicted by measuring the SEU rates in static random access memories (SRAMs). Martin Buehler of JPL has developed 4k SRAMs [*Buehler et al.*, 1990] that have a variable threshold for SEUs. Two of these SRAMs, one sensitive directly to the LET from protons and the other sensitive only to particles with $Z>1$, give an indication of the SEU rates from these two sources. An interesting CRRES result is that protons are a bigger contributor to SEUs than the $Z>1$ particles (M. Buehler, private communication, 1991). Although the process by which protons produce SEUs (nuclear interaction) has a much smaller cross section than from $Z>1$, there are so many more protons available that this is usually the main contributor.

The lower threshold channel measures protons directly without the need for nuclear interaction. About 10^{-8} of these protons produce a nuclear reaction. Since the sensitive area of this SRAM is about 2×10^{-6} m^2, it predicts the SEU rate of about 200 m^2 of sensitive area in normal circuits (where a nuclear reaction is required for upset). For the upper threshold channel that measures $Z>1$ particles, we get no such factor advantage. If we wish to get a sensitive area comparable to the whole spacecraft area, we would need to increase the number of chips.

While not quite as compact as the dose measuring FETs, the SRAMs still are small enough (28 pin IC) to be remotely placed near sensitive areas of the spacecraft if desired. The sensitive SRAMs and the RADFETs can be combined into a single chip or they can be kept separate. Mounting should be behind shieldings representative of the spacecraft. A baseline thickness is 0.040" (1 mm) and 0.150" (4 mm) of aluminum.

2. *S/C Charging Inputs*

A plug-in module for GEO or Polar orbits measures particles in appropriate energy bands for deep dielectric charging. These are particles able to penetrate roughly 5, 25, 250, and 2500 μm of aluminum, with perhaps the 5 μm thickness too low. Figure 2 shows the range of electrons and protons in plastic and aluminum. Most of the particles can be measured by solid state detectors behind absorbers. Pulse height thresholds and coincidence determine whether electrons or protons are being measured. One or more two element solid state telescopes are probably most appropriate The front detector measures low energy particles and acts as an absorber for the second element which responds to higher energies important for deep dielectric charging. Thinly shielded gas-filled tubes (Geiger tubes) are another possibility.

Solid state detectors do not readily measure electrons below something like 30 keV. If it is necessary to measure these, an electrostatic analyzer is appropriate. These are sufficiently complicated (high voltage, exposed detector) and bulky that it is an advantage to avoid their use. All of these detectors trigger at one to three flux thresholds to warn of possible charging.

A photodiode measures the presence of sunlight.

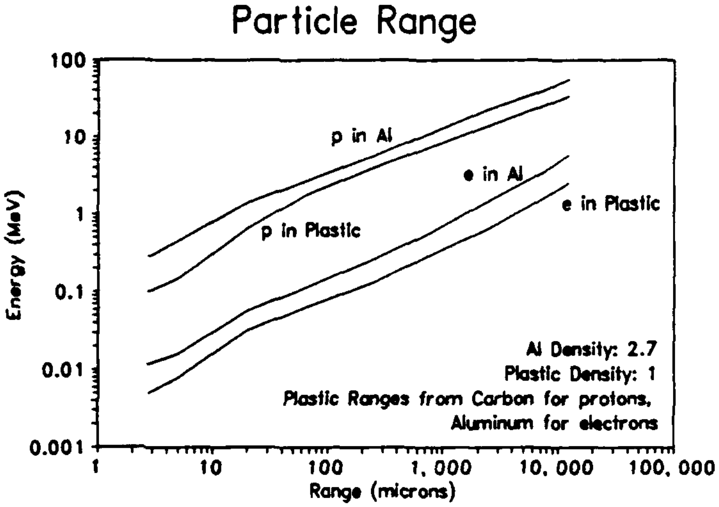

Figure 2. Range of protons and electrons in aluminum and plastic. The ranges for plastic are from the same column density in carbon and aluminum.

In the magnetosphere, conditions for negative body charging occur in the aurora but are most likely in GEO. The conditions are a hot electron spectrum and the absence of lower energy electrons (< 1 keV) that produce secondaries efficiently or of sunlight produced photoelectrons to balance the incoming current. Electrons above some 10 keV and adequate flux are needed. Actual body charging is indicated by the ion spectrum, which for the usual negative charging is absent low energy ions. This condition occurs very rapidly and is not generally hazardous. We therefore suggest that kilovolt ions need not be observed (it would require an ESA), and that 5-30 keV electrons need not be measured merely to sense body charging. The conditions which produce body charging can produce differential charging.

We need to examine the CRRES results to determine the optimum energy ranges and the significance of various types of charging.

3. Contamination

Another plug-in, it measures the total deposition, thermal surface degradation, and optical surface health. The remote sensors are put on or near the surface of interest. These sensors can be mixed or matched as appropriate for a particular application.

Mass Deposition

Mass Deposition as measured by a quartz crystal microbalance – physically a cylinder of 1"D x 3", these measure contaminates that condense on the instrument. The end of cylinder needs to look out into space. Weight is 120 g, power 350 mW. Sensitivity is about 1.5×10^{-9} g/cm^2, which corresponds to one Hz change in output signal. We recommend using warning thresholds at half the levels in : 0.5, 5.0, and 25×10^{-7} g/cm^2, and converting to thicknesses by assuming unit density.

Thermal Properties

Thermal Coating Calorimeter - A cylinder 1¼"D x 1", these measure the reflectivity change of whatever coating is put on the end of the cylinder. The most appropriate place for these is on or near the thermal radiator. They weigh 50 g and take 100 mW. We recommend this sensor to measure temperature changes corresponding to about 5×10^{-7} g/cm^2.

Optical Properties

Optical Surface Refraction as measured by a Bidirectional Reflective Distribution Function (BRDF) Sensor, a cylinder 1¼"D x 2". These measure the transmission of an optical surface. These would be placed near the optical systems like ATP. They weigh 100 g and take about 500 mW. This sensor would only be used in special circumstances.

S/C Charging Consequences

Another plug-in would be a transient pulse monitor and surface potential monitor for measuring charging consequences. This module would sense charging problems if they occur. It may not be appropriate for development during the first phase of the CEASE project.

Architecture

Mechanical

The basic package of processor plus Dose/SEU sensor could be built on two or three circuit boards. The sensitive elements must be behind known amounts of shielding and, therefore, must be near the outside of the spacecraft. Each plug-in module adds additional cards as necessary plus its appropriate sensors, which must be located with apertures that view out into space. Physically separating the sensors from the central unit means that only the sensors must be located on the outside. Integration of the instrument is more involved because additional units must be mounted and additional cables must be run from the sensors to the central unit. Integrating the sensors into the central unit means that space must be found on the outside of the spacecraft for a much larger volume.

The argument for modularity simplifies the development and could optimize the deployment of the diagnostics to regions of interest. However, there is a weight and complexity penalty for this approach.

If weight and size are the principal priority for the success of CEASE, then we feel that the instrument can be built as a single compact module that weighs significantly less than the SOW specified 15 pounds. The diagnostics' thresholds would be adjustable to accommodate particular spacecraft placement, orbit, and types of anomaly sensitivities. An instrument's "personality" would be customized via ROM and TBD component values. The spacecraft interface to CEASE would also be part of this "personality" definition.

Electronics

For breadboard purposes, we recommend a 80C85 microprocessor for the central processor. These can be bought relatively inexpensively off the shelf. We have experience programming these and believe that this is a cost effective development route. When the breadboard testing and evaluation is finished, we will have the input we need to develop a more specialized or radiation-hard processor if necessary. Efficient coding of the 80C85 may demonstrate its suitability for the CEASE.

Most of the other specialized electronics for these types of detectors are fairly well known. Size, power, and packaging would be the principal concerns. Stable and simple threshold adjustment strategies will be developed.

Software and Algorithms

Software and algorithms will be developed, tested, and evaluated during the breadboard phase. Particular attention will be paid to issues of calibration and testing for the engineering unit. Flight-model CEASE devices will have only limited data output since they are engineering devices rather than scientific research hardware. The development unit will need to be a little of both in order to demonstrate its capabilities.

One way of doing this is to archive in CEASE memory a limited comprehensive record of recent diagnostic outputs and status. This record would be continuously written into a FIFO style buffer. A special interrogation to the CEASE would empty the buffer providing a "snapshot" of a single time interval. If this buffer were interrogated frequently enough, a near real time data stream would result. This same record would contain historic summations of some diagnostic; for example, the total accumulated radiation dosage or optical contamination. These factors would need to be saved in non-volatile memory in a redundant fashion to insure their retention during power down.

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